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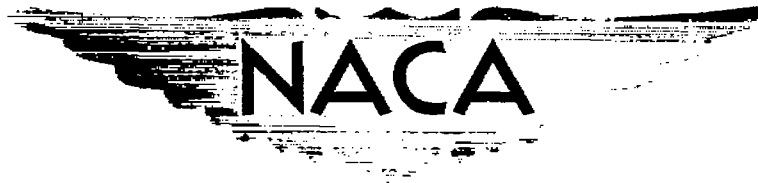
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RESEARCH MEMORANDUM

LATERAL-CONTROL INVESTIGATION OF FLAP-TYPE CONTROLS ON
A WING WITH QUARTER-CHORD LINE SWEPT BACK 60° , ASPECT
RATIO 4, TAPER RATIO 0.6, AND NACA 65A006 AIRFOIL SECTION

TRANSONIC -BUMP METHOD

By Raymond D. Vogler

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RESEARCH MEMORANDUM

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SUMMARY

As part of an NACA research program, an investigation by the transonic-bump method through a Mach range of 0.7 to 1.15 has been made in the Langley high-speed 7- by 10-foot tunnel to determine the lateral-control characteristics of 30-percent-chord flap-type controls of various spans and locations. The wing of the semispan fuselage-wing combination had 60° of sweepback of the quarter-chord line, a taper ratio of 0.6, an aspect ratio of 4.0, and an NACA 65A006 airfoil section parallel to the free air stream.

Rolling and pitching moments and lift data were obtained through a small range of control deflections. The data are presented as control-effectiveness parameters to show their variation with Mach number. A moderate and gradual decrease in aileron and lift effectiveness occurred with increase in Mach number starting at a Mach number of approximately 0.9. Little variation in pitching effectiveness with Mach number occurred for the outboard controls below a Mach number of 1.0; above 1.0 considerable loss occurred except for the short-span control at the wing tip.

INTRODUCTION

The need for aerodynamic data in the transonic speed range and the fact that such data are lacking or incomplete have led to the establishment by the NACA of an integrated program of transonic research. As a part of the transonic research program, a series of wing-fuselage configurations having wing plan form as the chief variable are being investigated in the Langley high-speed 7- by 10-foot tunnel by using the transonic-bump test method.

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This paper presents the results of a lateral-control investigation of a semispan wing-fuselage model employing a wing with the quarter-chord line swept back 60° , an aspect ratio of 4, a taper ratio of 0.6, and an NACA 65A006 airfoil section. The purpose of this investigation was to obtain lateral-control data with flap-type controls of 30-percent chord and various spans. The results of a previous investigation of the same wing-fuselage without controls, giving additional aerodynamic data, may be found in reference 1. Previous lateral-control data published in this series are presented in references 2 and 3.

MODEL AND APPARATUS

The semispan wing had 60° of sweepback at the quarter-chord line, an aspect ratio of 4, a taper ratio of 0.6, and an NACA 65A006 airfoil section (reference 4) parallel to the free air stream (fig. 1). The wing was made of steel and the fuselage was made of brass. The wing was mounted in the center of the fuselage vertically and had no dihedral or incidence. The regular transonic-research fuselage (reference 2), semicircular in cross section, was bent to the contour of the bump.

The controls (aileron or flap) were made integral with the wing by cutting grooves 0.03 inch wide along the 70-percent-chord line on the upper and lower surfaces of the wing (fig. 2). After setting the control at the desired deflection by bending the metal along the grooves, the grooves were filled with wax, thus giving a close approach to a 30-percent-chord sealed plain flap-type control surface. The entire control span from fuselage surface to wing tip was divided into four equal spanwise segments.

The model was mounted on an electrical strain-gage balance wired to a calibrated potentiometer in order to measure the aerodynamic forces and moments. The balance was mounted in a chamber within the bump, and the chamber was sealed except for a small hole through which an extension of the wing passed. This hole was covered by the fuselage and plate which was approximately 0.06 inch above the bump surface.

COEFFICIENTS AND SYMBOLS

C_L lift coefficient $\left(\frac{\text{Twice lift of semispan model}}{qS} \right)$

C_{l_a} rolling-moment coefficient produced by the control (rolling-moment coefficient with control deflected minus rolling-moment coefficient without deflection). Rolling-moment coefficient at plane of symmetry equals rolling moment of semispan model divided by qSb .

- C_m pitching-moment coefficient referred to $0.25\bar{c}$
 $\left(\frac{\text{Twice pitching moment of semispan model}}{qS\bar{c}} \right)$
- q effective dynamic pressure over span of model, pounds per square foot $\left(\frac{1}{2} \rho V^2 \right)$
- S twice wing area of semispan model, 0.125 square foot
- b twice span of semispan model, 0.707 foot
- \bar{c} mean aerodynamic chord of wing, 0.1805 foot $\left(\frac{2}{S} \int_0^{b/2} c^2 dy \right)$
- c local wing chord, feet
- y spanwise distance from plane of symmetry
- y_1 spanwise distance from plane of symmetry to inboard end of control
- ρ mass density of air, slugs per cubic foot
- V free-stream air velocity, feet per second
- M effective Mach number over span of model $\left(\frac{2}{S} \int_0^{b/2} c M_a dy \right)$
- M_a average chordwise local Mach number
- M_l local Mach number
- R Reynolds number of wing based on \bar{c}
- α angle of attack of wing root chord line, degrees
- δ control deflection relative to wing-chord plane, measured in a plane perpendicular to control hinge axis (positive when trailing edge is down), degrees
- b_a control span, measured perpendicular to plane of symmetry
- $C_{L\delta} = \left(\frac{\partial C_L}{\partial \delta} \right)_\alpha$ (lift-effectiveness parameter)
- $C_{l\delta} = \left(\frac{\partial C_{l_a}}{\partial \delta} \right)_\alpha$ (aileron-effectiveness parameter)

$$C_{m\delta} = \left(\frac{\partial C_m}{\partial \delta} \right)_{\alpha} \text{ (pitching-effectiveness parameter)}$$

The subscript α indicates the factor held constant.

CORRECTIONS

The aileron-effectiveness parameters presented herein represent the aerodynamic effects on a complete wing produced by the deflection of the control on only one semispan of the complete wing. Reflection-plane correction factors as given in figure 3 have been applied to the parameters throughout the Mach range of the investigation. The values of the correction factors were obtained from unpublished experimental low-speed data and theoretical considerations. Although the corrections are based on low-speed considerations and are valid for low Mach numbers only (probably too large at high Mach numbers), it is believed that the results obtained by applying the corrections give better representation of true conditions than uncorrected data. No attempt has been made to correct the rolling-moment data for increments of rolling moment caused by asymmetrical pressure distribution on the end plate as a result of control deflection. This effect is believed to be of little significance for short-span outboard control surfaces but may be of importance for control surfaces that extend outboard from the wing-fuselage intersection.

The lift-effectiveness and pitching-effectiveness parameters represent the aerodynamic effects of deflection in the same direction of the controls on both semispans of the complete wing. No reflection-plane corrections are necessary for the lift and pitching-moment data.

No corrections were applied for any twisting of the wing or deflection of the controls caused by the air load. Static load tests indicated that such twisting or deflection was negligible.

TEST TECHNIQUE

The investigation was made in the Langley high-speed 7- by 10-foot tunnel using an adaptation of the NACA wing-flow technique for obtaining transonic speeds. The technique used involves placing the model in the high-velocity flow field generated over the curved surface of a bump on the tunnel floor (reference 5). Typical contours of local Mach number in the vicinity of the model location on the bump with model removed are shown in figure 4. The contours indicate that there is a Mach number variation of about 0.04 over the wing semispan at low Mach numbers and about 0.07 at high Mach numbers. The chordwise Mach number variation

is generally less than 0.01. No attempt has been made to evaluate the effects of this chordwise and spanwise Mach number variation. The long-dash lines near the root of the wing in figure 4 indicate a local Mach number 5 percent below the maximum value and represent the estimated extent of the bump boundary layer. The effective test Mach number was obtained from contour charts similar to those presented in figure 4 by using the relationship

$$M = \frac{2}{S} \int_0^{b/2} cM_a dy$$

The variation of the mean test Reynolds number with Mach number is shown in figure 5.

Force and moment data were obtained with controls of various spans through a Mach number range of 0.70 to 1.15, an angle-of-attack range of -6° to 6° , and control deflections of 0° , 5° , and 10° . Some rolling-moment data were obtained on the 43-percent-span outboard control at a deflection of 2° .

RESULTS AND DISCUSSION

In figures 6 to 9 are curves of lift, rolling-, and pitching-moment coefficients plotted against control deflection for the 21-percent-span outboard, the 43-percent-span outboard, the 86-percent-span outboard, and the 43-percent-span inboard controls at a wing angle of attack of 2° . Inasmuch as the wing was symmetrical, data obtained at negative angles of attack and positive control deflections were considered, with appropriate regard to signs, to be equivalent to data that would be obtained at positive angles of attack and negative control deflections and were plotted as such. The curves of figures 6 to 9 are typical of the curves of all the data obtained.

Control-effectiveness parameters. - The control-effectiveness parameters presented in figures 10 to 12 were obtained from figures 6 to 9 and similar plots of the data for the various control configurations and angles of attack. The variation of control effectiveness with control deflection was linear throughout the deflection range investigated ($\pm 10^\circ$) for all configurations.

Starting at a Mach number of approximately 0.9 (figs. 10 and 11), a moderate and gradual decrease in aileron and lift effectiveness occurs with increase in Mach number. This moderate loss in effectiveness in the transonic speed range for this highly swept wing is less pronounced than

the loss in effectiveness for wings with less sweep (references 2 and 3). Figure 11 indicates that the aileron effectiveness of the 43-percent-span inboard and the 21-percent-span outboard controls are less affected by Mach number than the 43-percent-span and the 86-percent-span outboard controls. However, since the rolling moments of the 21-percent-span outboard control were small, any variations with Mach number may have been masked by fluctuation of forces, especially at the higher Mach numbers.

The curves of pitching-moment parameters of the 43-percent-span and the 86-percent-span outboard controls (fig. 12) follow the same general loss in effectiveness trend with Mach number as the curves of the other parameters except that the loss in effectiveness starts at a Mach number of approximately 1.0. The data indicate that the 43-percent-span inboard and the 21-percent-span outboard controls are about equal in pitching effectiveness except above a Mach number of 0.95.

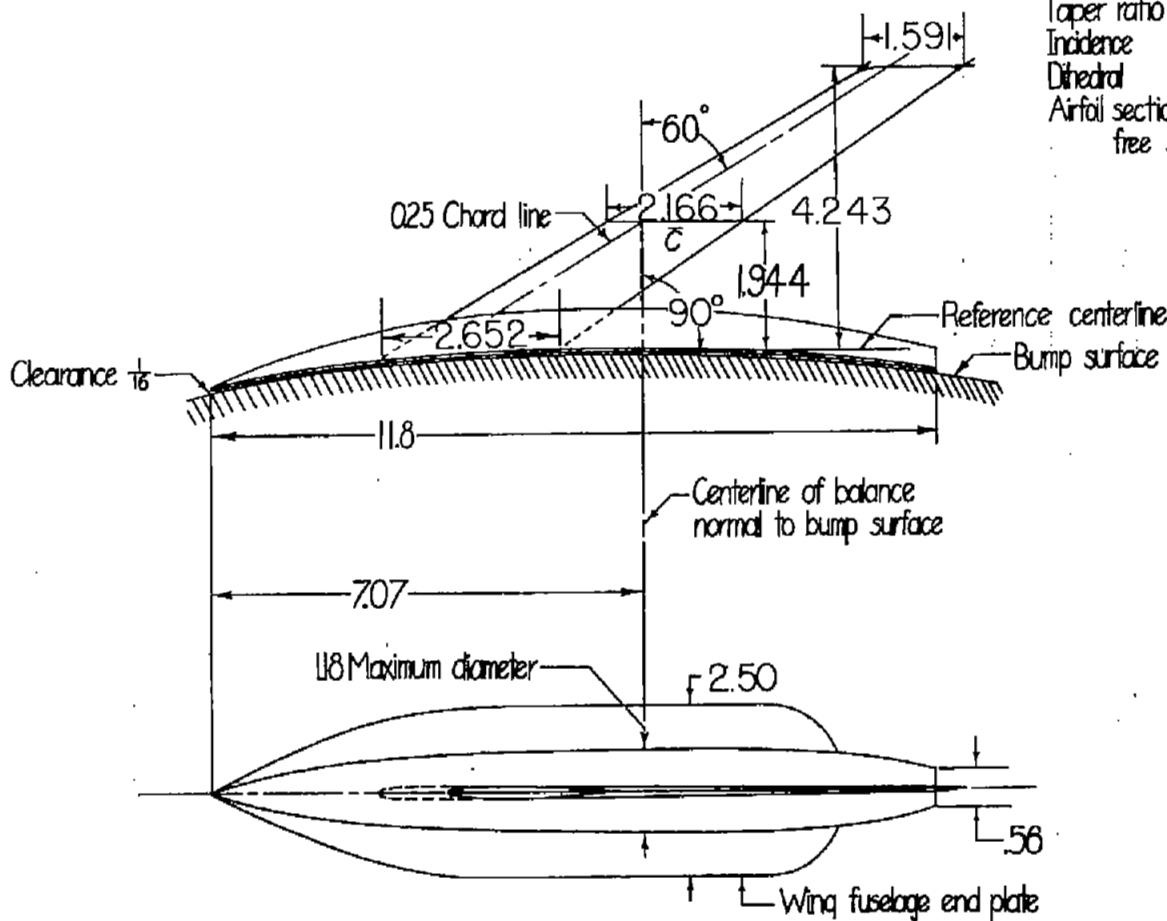
Figure 13 showing the effectiveness of controls of various spans starting at the wing tip indicates that the 21-percent-span outboard control gives low aileron effectiveness. The pitching effectiveness of the 21-percent-span outboard control, however, is better retained at Mach numbers of 1.0 and above.

A comparison of the values of $C_{l\delta}$ obtained at a Mach number of 0.7 in this investigation with those estimated by the method of reference 6 shows good agreement (fig. 14).

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1. King, Thomas J., Jr., and Myers, Boyd C., II: Aerodynamic Characteristics of a Wing with Quarter-Chord Line Swept Back 60° , Aspect Ratio 4, Taper Ratio 0.6, and NACA 65A006 Airfoil Section. Transonic-Bump Method. NACA RM L9G27, 1949.
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5. Schneider, Leslie E., and Ziff, Howard L.: Preliminary Investigation of Spoiler Lateral Control on a 42° Sweptback Wing at Transonic Speeds. NACA RM L7F19, 1947.
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Tabulated Wing Data	
Area (Twice semispan)	0.125 sq ft
Mean aerodynamic chord	0.1805 ft
Aspect ratio	4.0
Taper ratio	0.6
Incidence	0.0°
Dihedral	0.0°
Airfoil section parallel to free stream	NACA 65A006

All dimensions in inches.

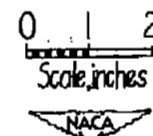


Figure 1.- General arrangement of model with 60° sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil.

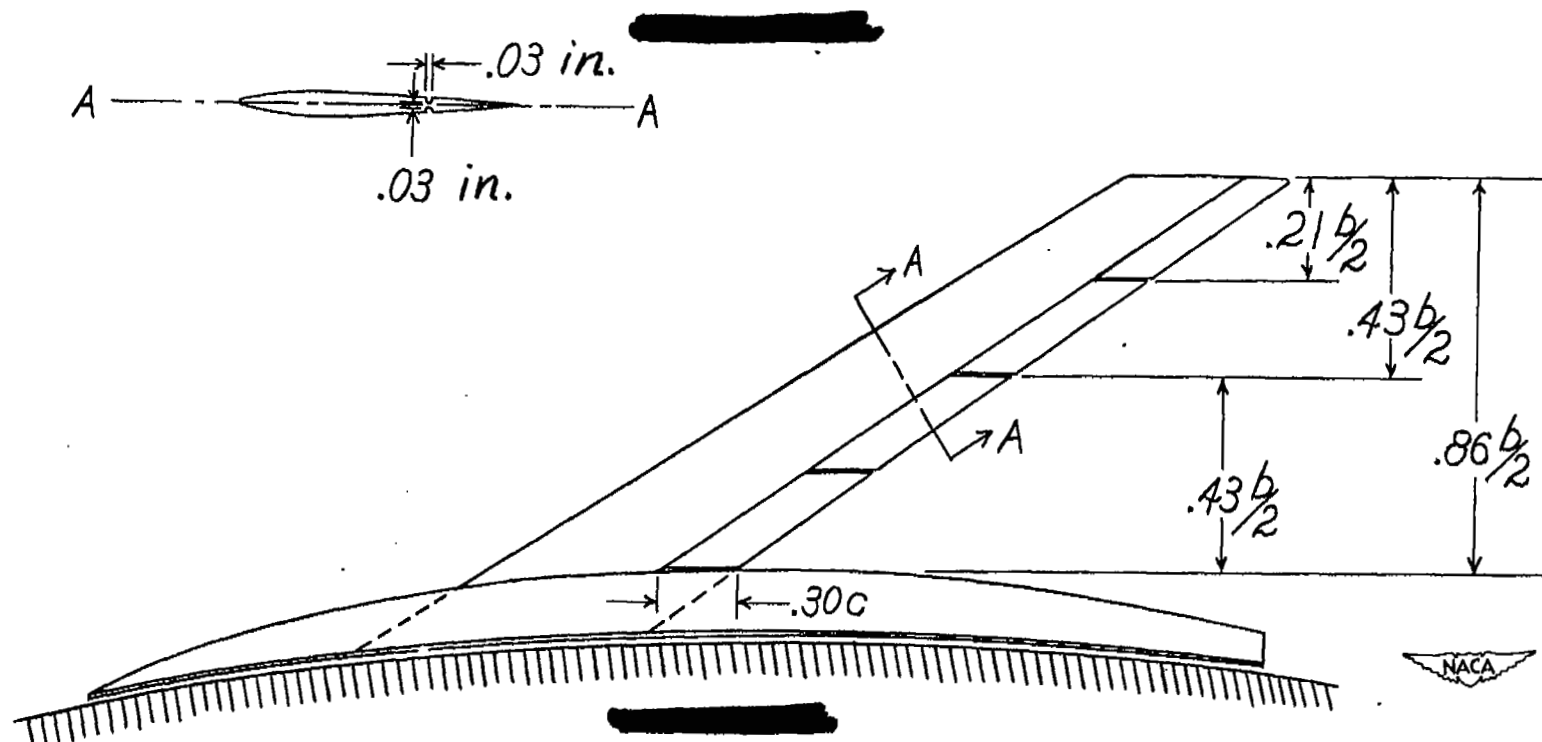


Figure 2.- Details of the controls.

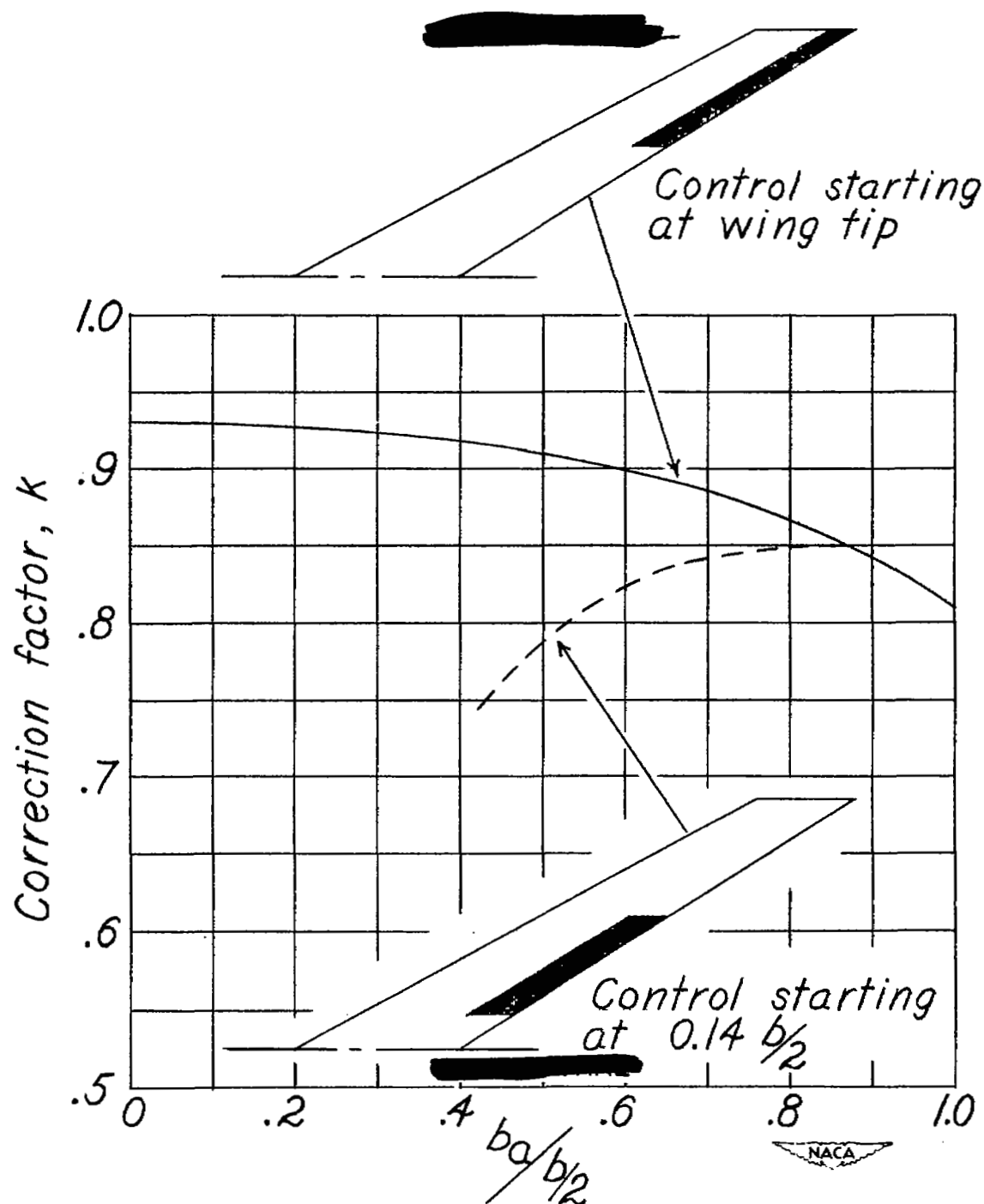


Figure 3.- Reflection-plane correction factors for inboard and outboard controls of various spans for a wing of 60° sweepback, aspect ratio 4, and taper ratio of 0.6.

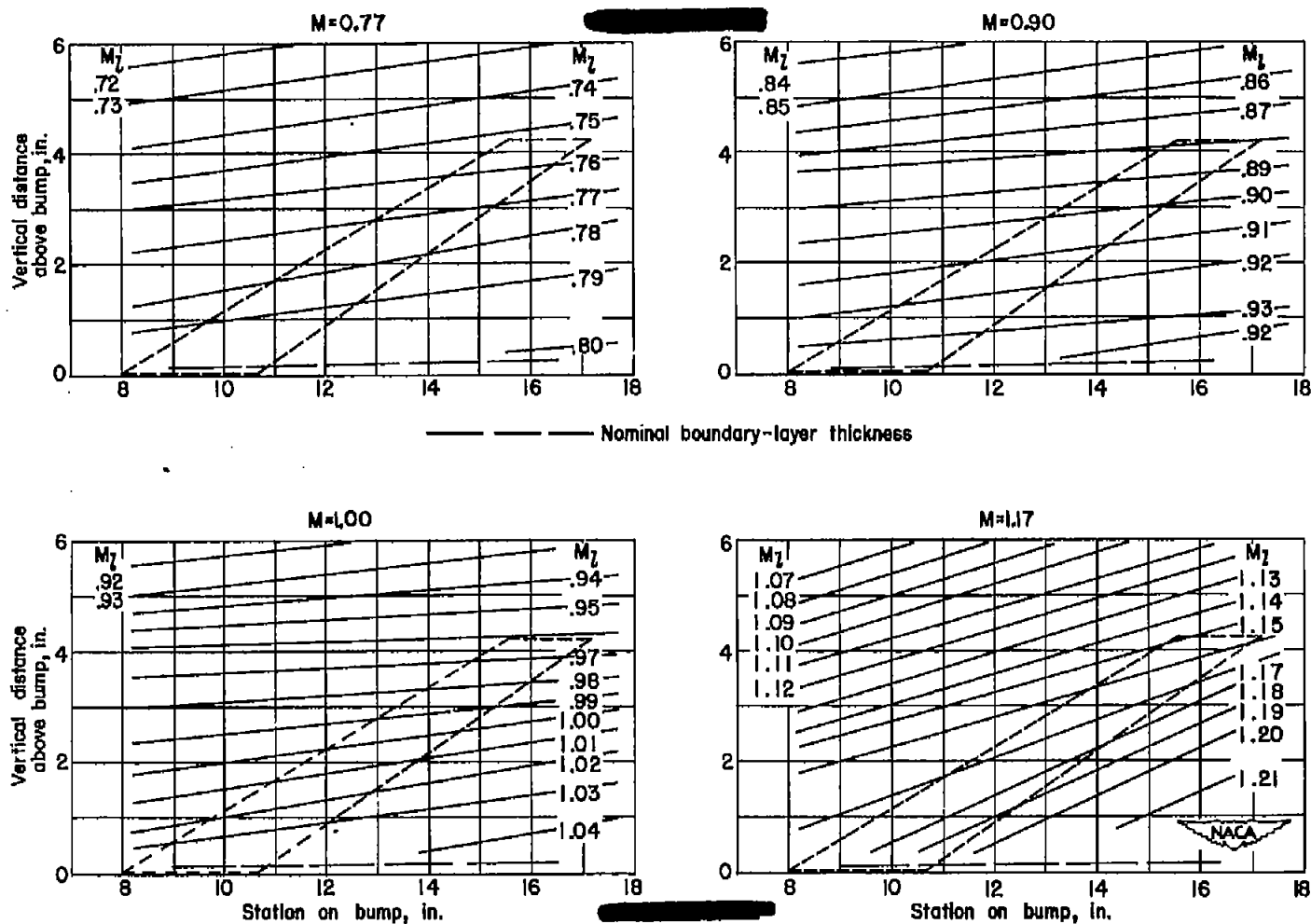


Figure 4.- Typical Mach number contours over transonic bump in region of model location.

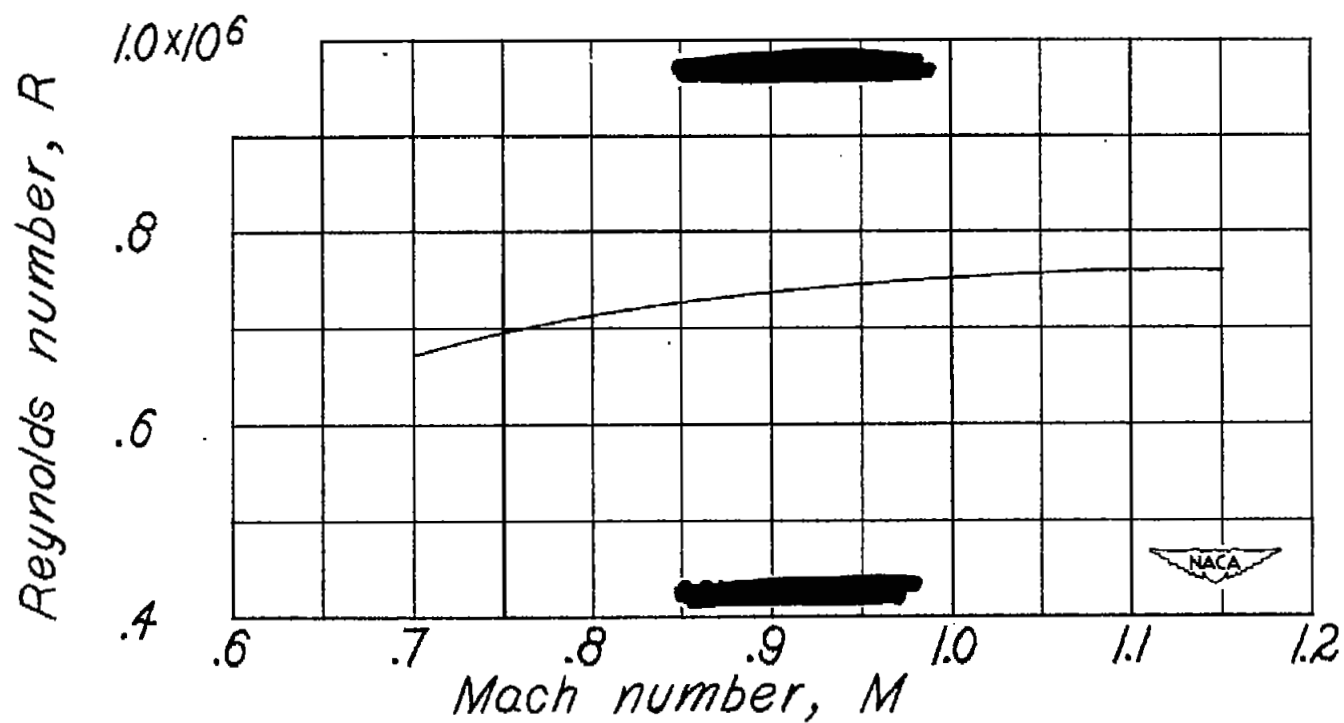


Figure 5.- Variation of mean test Reynolds number with Mach number for model with 60° sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil.

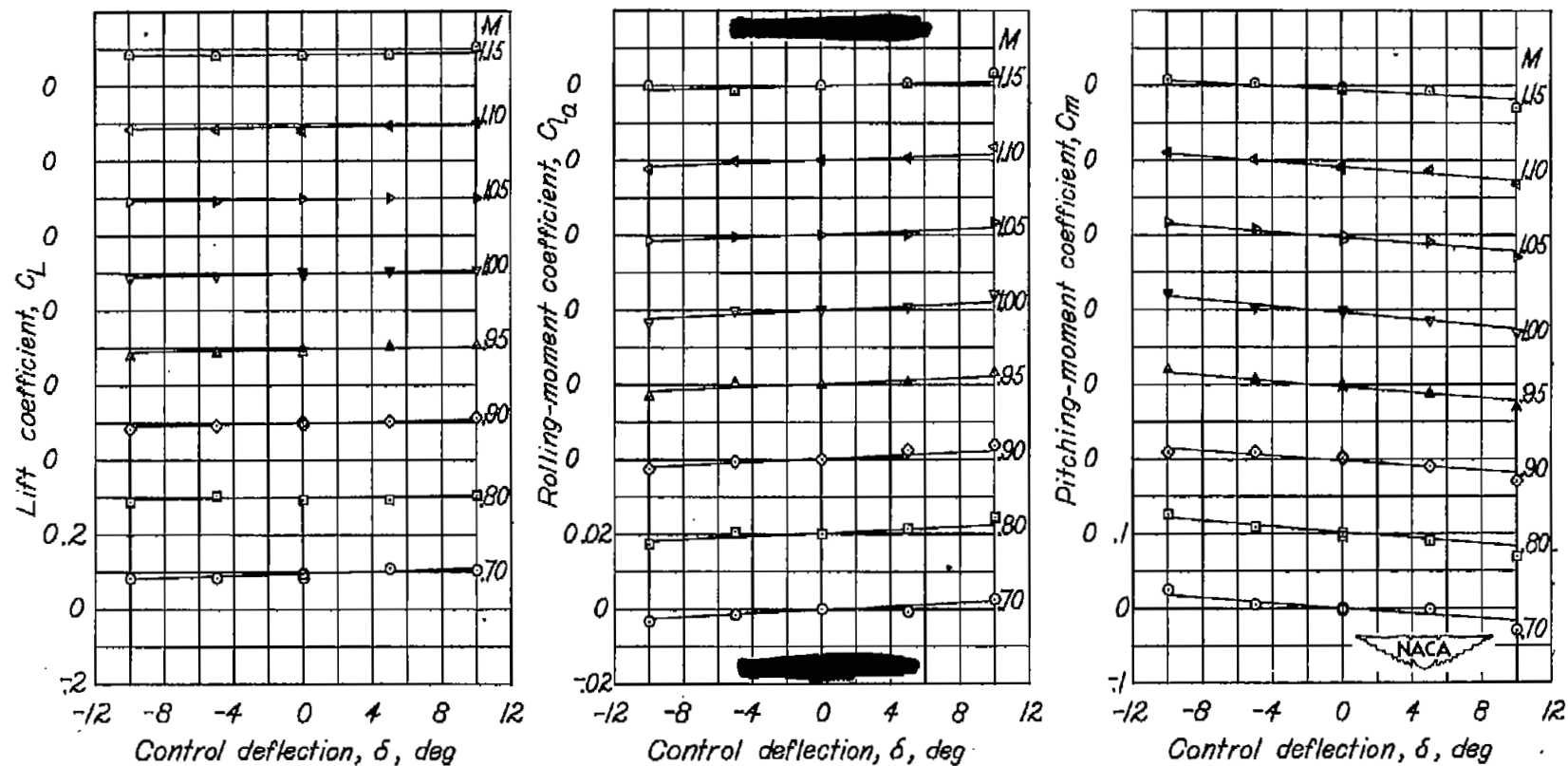


Figure 6.- Variation of lift, rolling-moment, and pitching-moment coefficients with control deflection for various Mach numbers. $b_a = 0.21 \frac{b}{2}$, outboard; $\alpha = 2^\circ$.

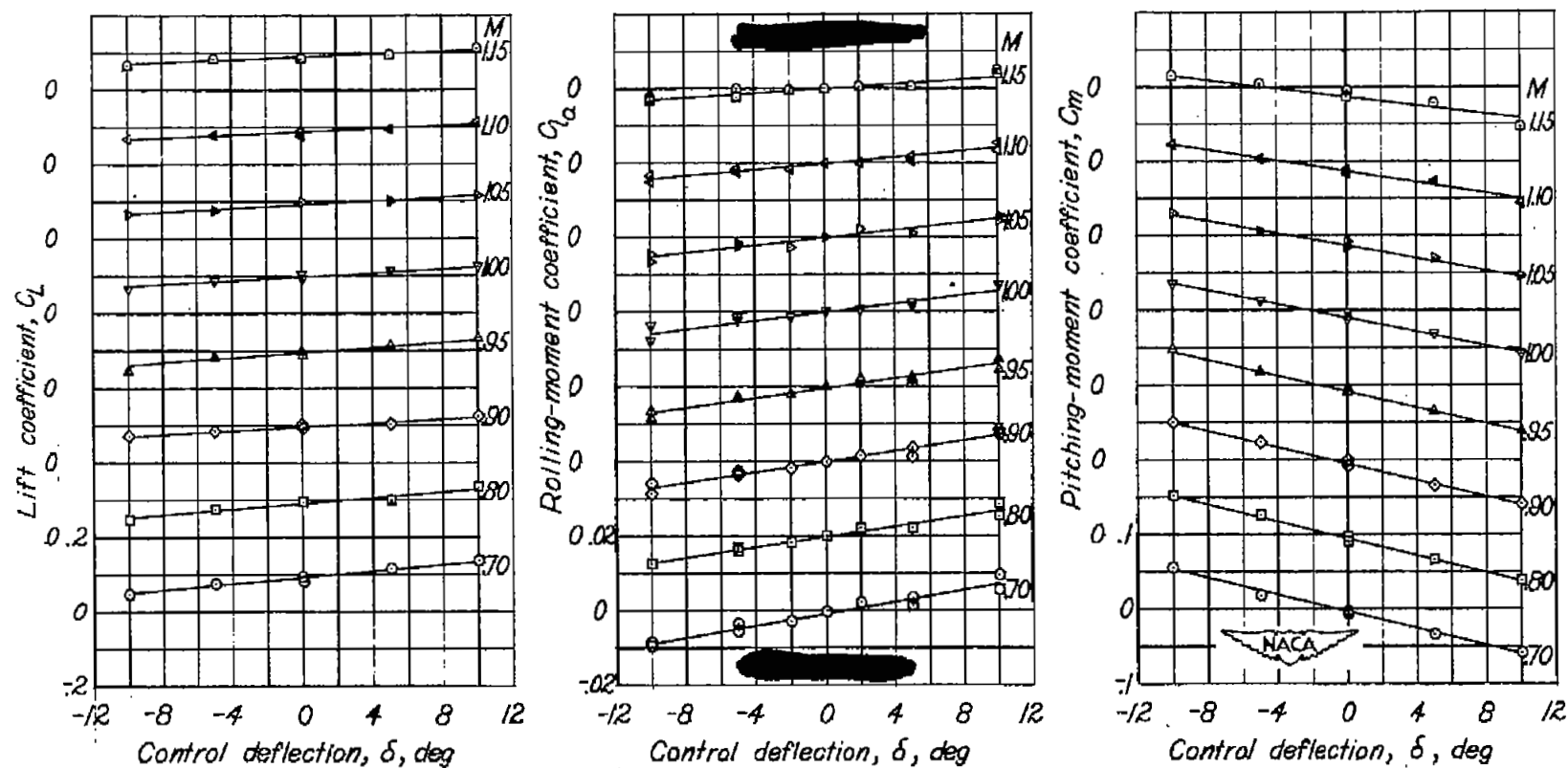


Figure 7.- Variation of lift, rolling-moment, and pitching-moment coefficients with control deflection for various Mach numbers. $b_a = 0.43 \frac{b}{2}$, outboard; $\alpha = 2^\circ$.

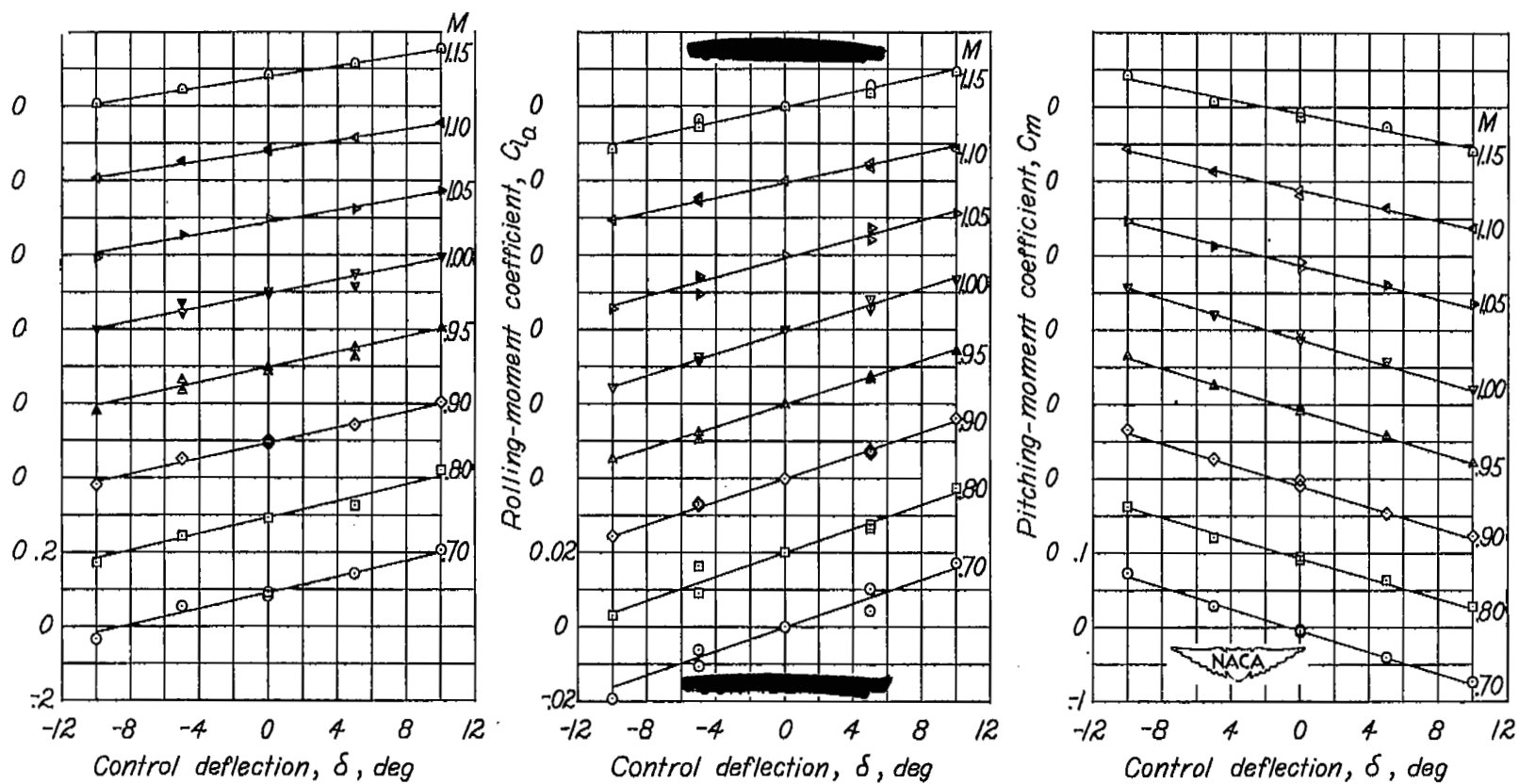


Figure 8.- Variation of lift, rolling-moment, and pitching-moment coefficients with control deflection for various Mach numbers. $b_a = 0.86 \frac{b}{2}$, outboard; $\alpha = 2^\circ$.

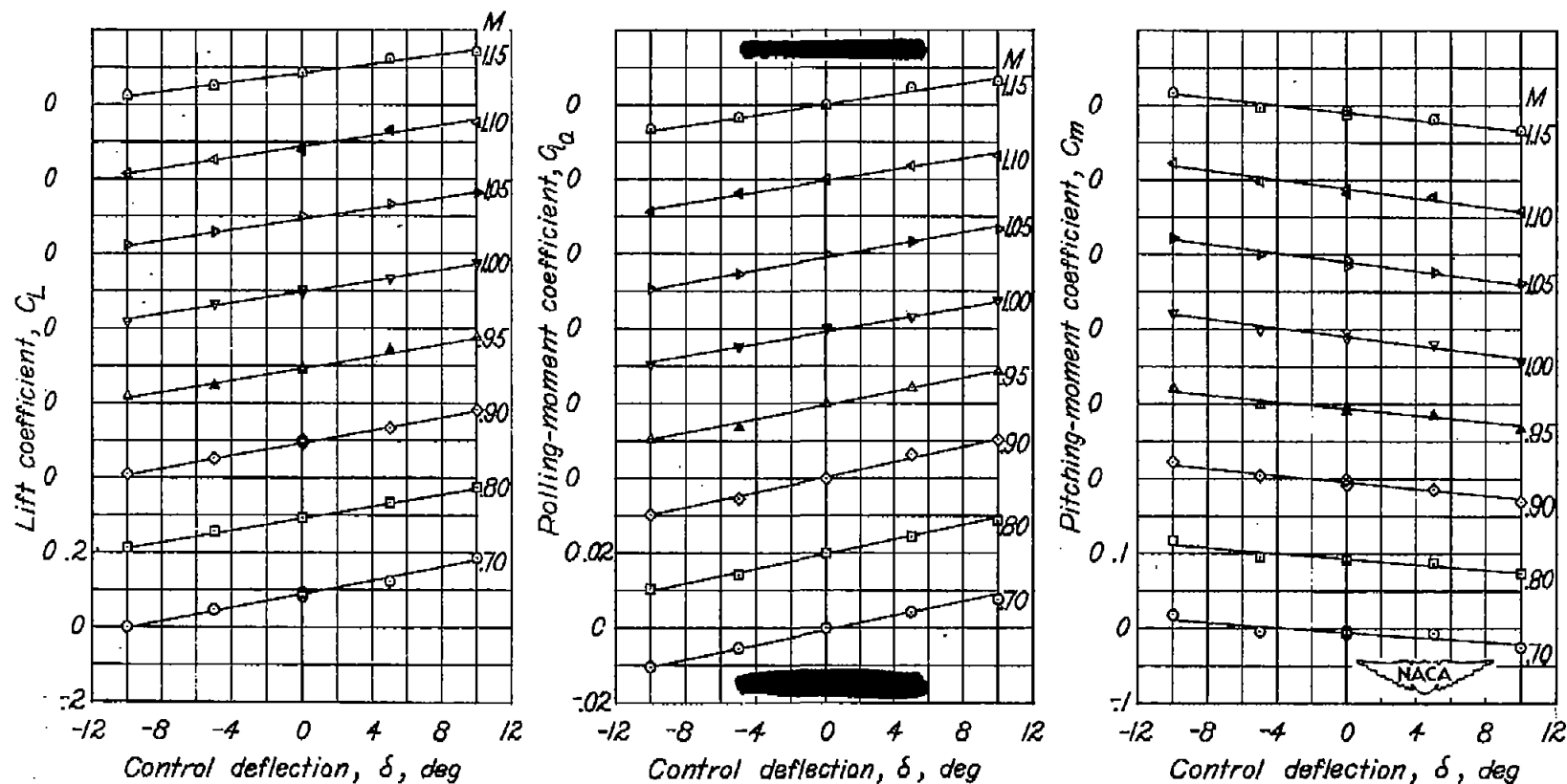


Figure 9.- Variation of lift, rolling-moment, and pitching-moment coefficients with control deflection for various Mach numbers. $b_a = 0.43 \frac{b}{2}$, inboard; $\alpha = 2^\circ$.

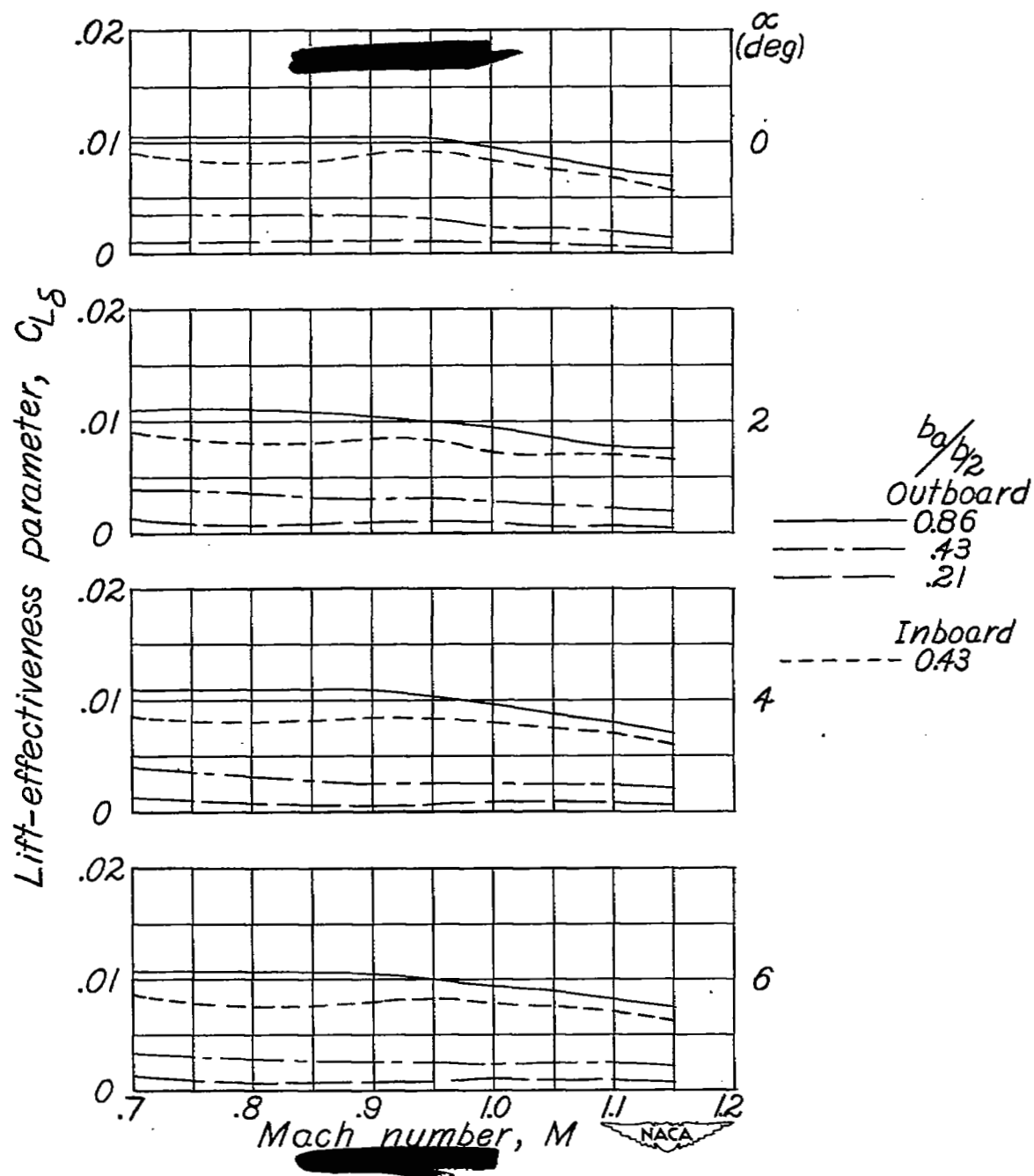


Figure 10.- Variation of lift-effectiveness parameter with Mach number.

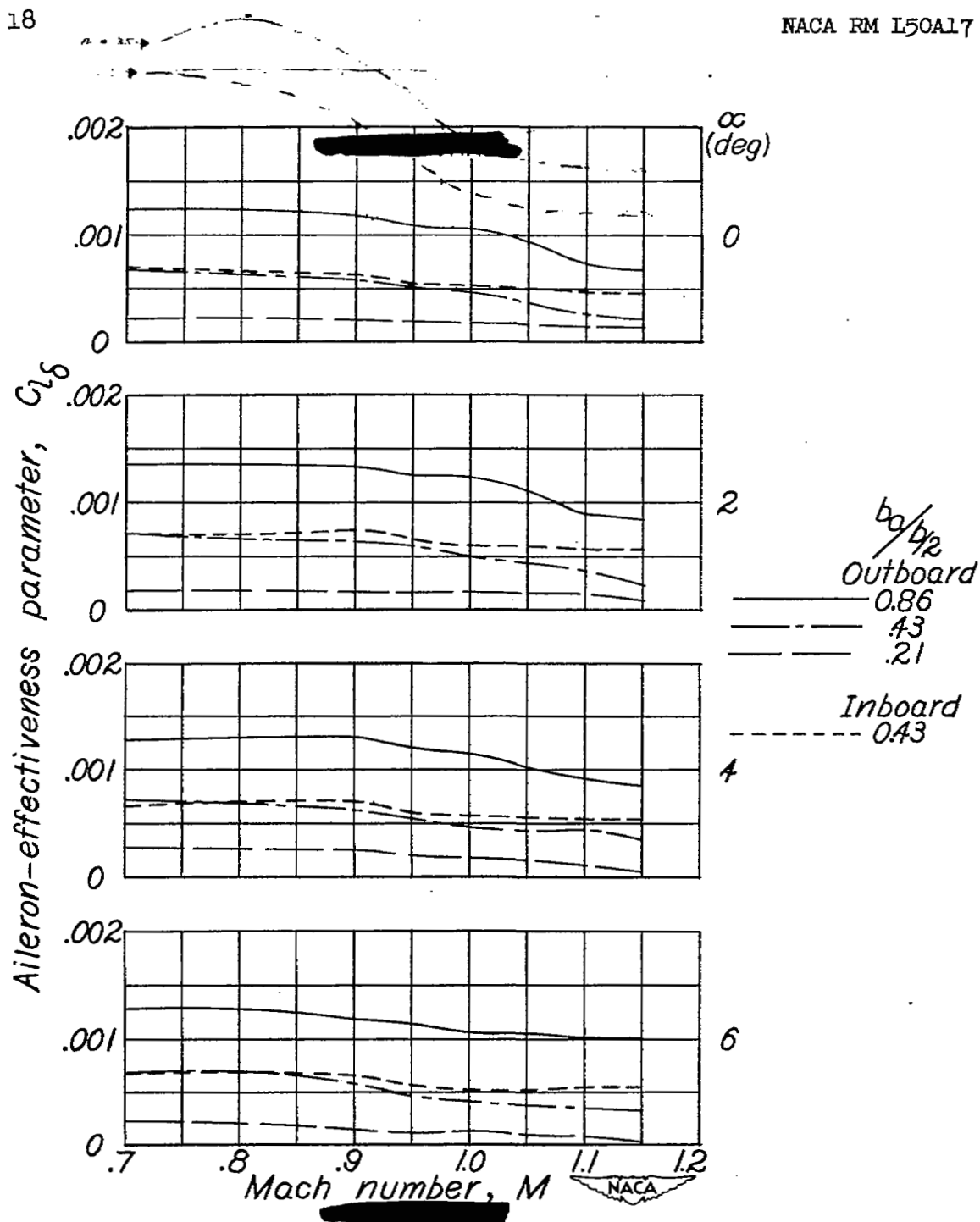
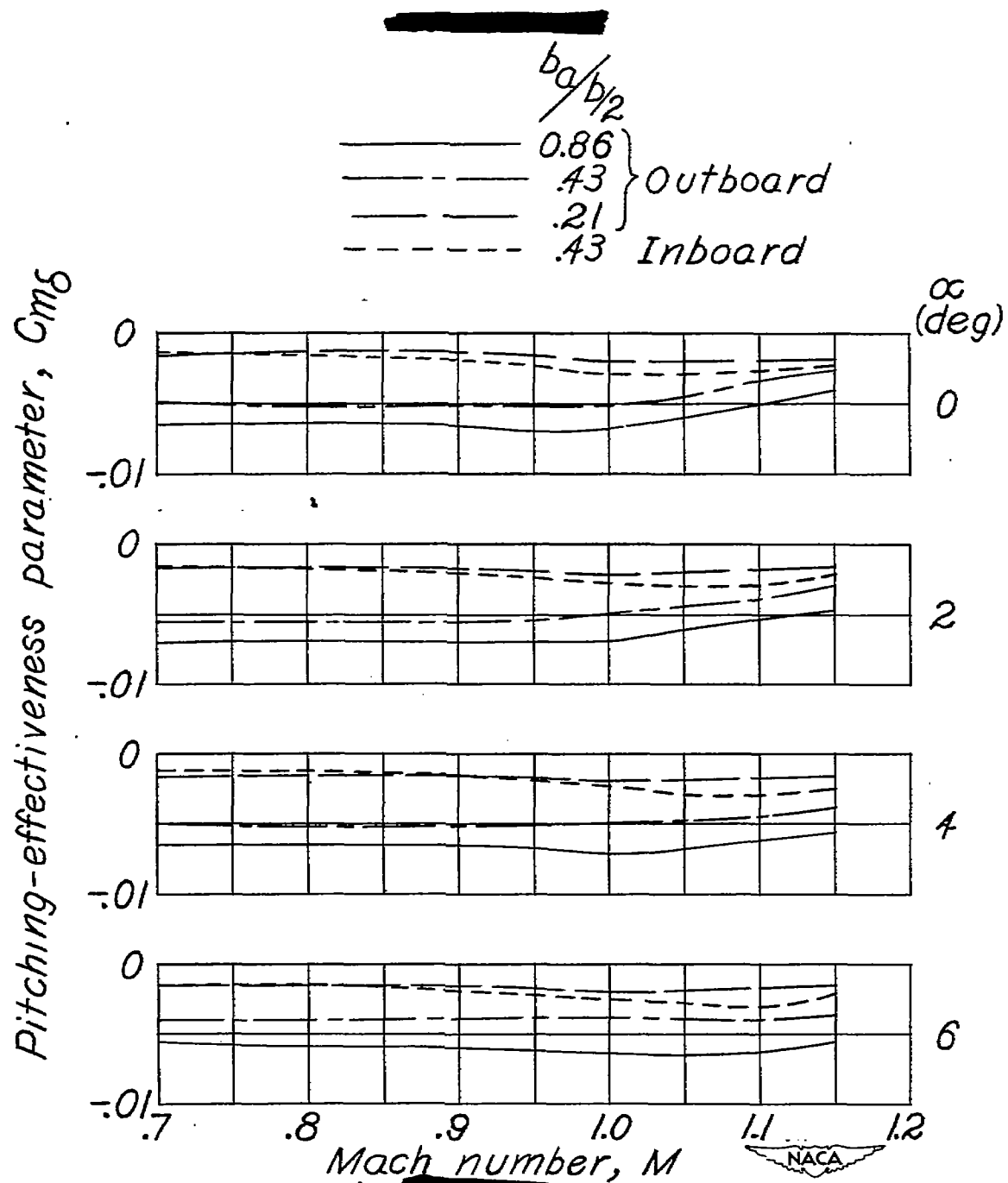


Figure 11.- Variation of aileron-effectiveness parameter with Mach number.



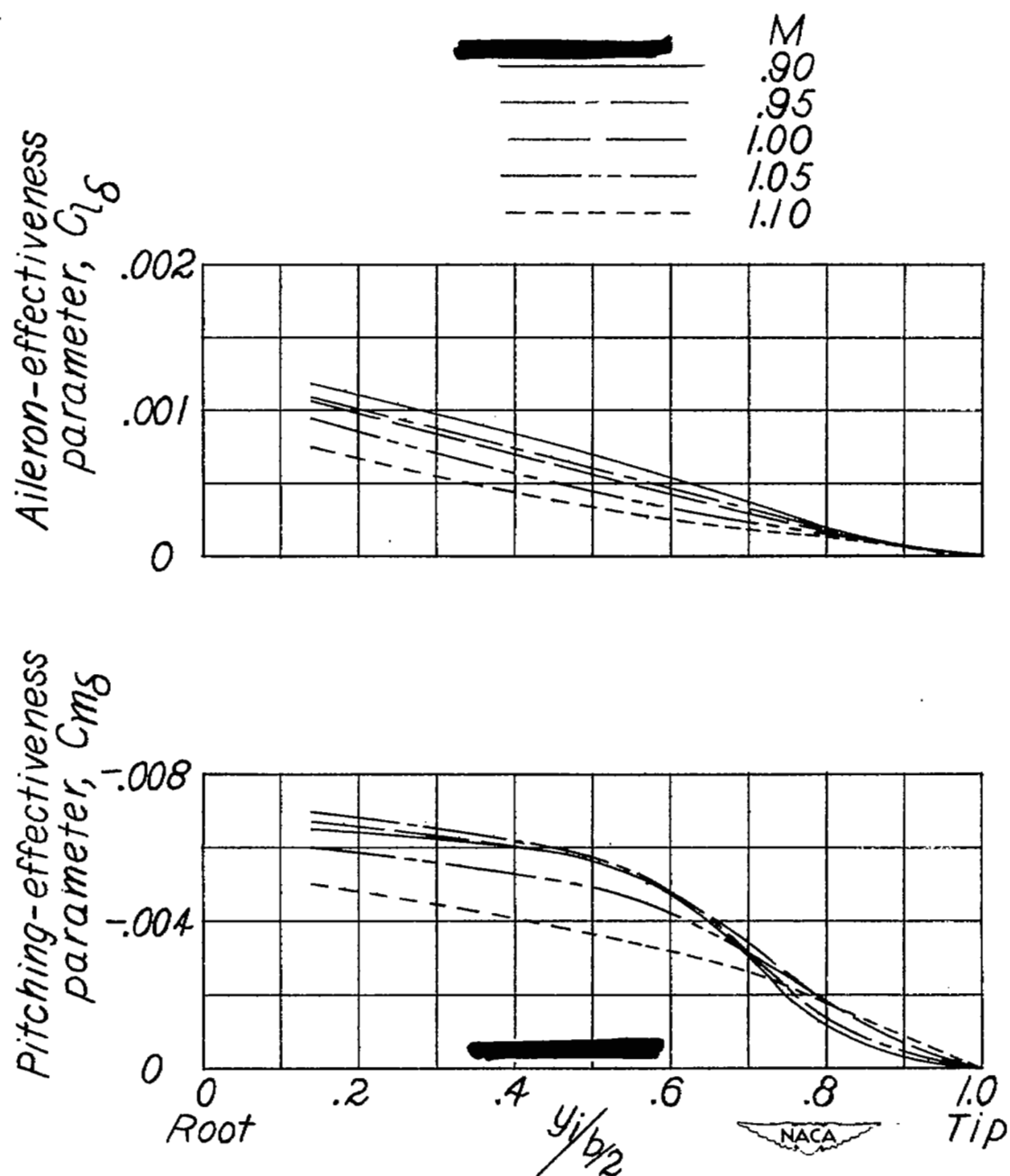


Figure 13.- Variation of control-effectiveness parameters with control span starting at the wing tip for various Mach numbers. $\alpha = 0^\circ$.

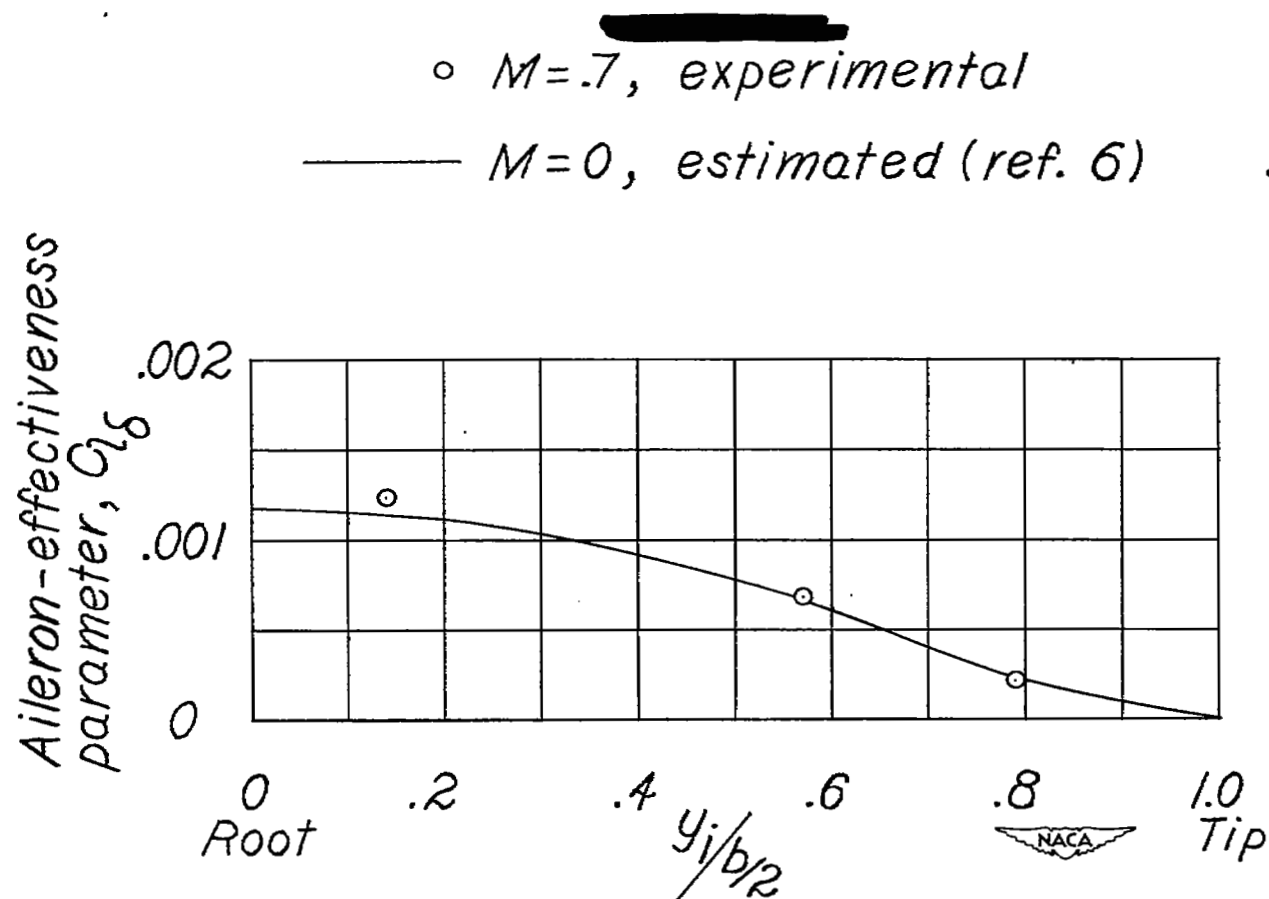


Figure 14.- Comparison of the experimental and estimated variation of aileron effectiveness with aileron span. $\alpha = 0^\circ$.

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